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Space Exploration Programs and Space Sciences

For the Period January 1 to February 28, 1967

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SPACE PROGRAMS SUMMARY 37-44, VOL. VI

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Preface

The Space Programs Summary is a six-volume bimonthly publication designed to report on JPL space exploration programs and related supporting research and advanced development projects. The titles of all volumes of the Space Programs Summary are:

Vol. I. *The Lunar Program* (Confidential)

Vol. II. *The Planetary–Interplanetary Program* (Confidential)

Vol. III. *The Deep Space Network* (Unclassified)

Vol. IV. *Supporting Research and Advanced Development* (Unclassified)

Vol. V. *Supporting Research and Advanced Development* (Confidential)

Vol. VI. *Space Exploration Programs and Space Sciences* (Unclassified)

The Space Programs Summary, Vol. VI, consists of: an unclassified digest of appropriate material from Vols. I, II, and III; an original presentation of the JPL quality assurance and reliability efforts, and the environmental- and dynamic-testing facility-development activities; and a reprint of the space science instrumentation studies of Vols. I and II. This instrumentation work is conducted by the JPL Space Sciences Division and also by individuals of various colleges, universities, and other organizations. All such projects are supported by the Laboratory and are concerned with the development of instruments for use in the NASA space flight programs.

Approved by:

W. H. Pickering, *Director*

Jet Propulsion Laboratory

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I. *Surveyor* Project

THE LUNAR PROGRAM

N67-27742

A. Introduction

The *Surveyor* flight spacecraft are designed to span the gap between the *Ranger* Project and the *Apollo* Project by making soft landings on the Moon to extend our knowledge of lunar conditions and determine the suitability of sites for proposed *Apollo* spacecraft landings. Hughes Aircraft Company (HAC), Space Systems Division, is under contract to fabricate the *Surveyor A-21* spacecraft. The launch vehicle, a combination *Atlas/Centaur*, is provided by General Dynamics/Convair. Control, command, and tracking functions for the *Surveyor* missions are performed by the *Surveyor* Mission Operations System and the JPL Deep Space Network.

Surveyor I, the first flight spacecraft, was launched from Cape Kennedy, Florida, on May 30, 1966, and soft-landed on the Moon on June 2, 1966. By June 14, when lunar sunset occurred, approximately 100,000 commands had been received by the spacecraft, and 10,338 pictures of the spacecraft and its immediate vicinity had been transmitted. During the second lunar day, 812 additional pictures were transmitted by *Surveyor I*.

Surveyor II, the second flight spacecraft, was launched from Cape Kennedy on September 20, 1966. The spacecraft performed nominally until the command was given for midcourse thrust execution. At that time, one vernier engine did not ignite. The resulting imbalance of thrust from the other two vernier engines imposed a tumbling motion on the spacecraft, from which it failed to recover. The *Surveyor II* mission was terminated on September 22 after all contact with the spacecraft had been lost.

The third *Surveyor* flight spacecraft is scheduled for launch during April 1967.

B. Systems Engineering and Testing

1. SC-3 (Third Flight Spacecraft)

In December 1966 the SC-3 spacecraft was returned to the Hughes Aircraft Company from the Eastern Test Range for reverification tests. The vernier engine 2 heater lines were modified and the roll attitude jet attach bracket was redesigned. (The latter change was also incorporated in SC-4 through -7.) The SC-3 spacecraft was then shipped to the Eastern Test Range on February 12, 1967. First mate with the *Centaur* vehicle is planned for March 4, followed by the joint flight acceptance composite test on March 7 and demate on March 8, 1967. Final mate is planned for April 11, 1967.

2. SC-4 (Fourth Flight Spacecraft)

Because of SC-3 reverification tests, the SC-4 spacecraft was removed from solar-thermal-vacuum testing before completion of the last phase. Plans call for completion of solar-thermal-vacuum and vibration testing by mid-April, when the spacecraft will be given the combined systems test.

3. SC-5 Through -7 (Fifth Through Seventh Flight Spacecraft)

The payload upgrade period for SC-5 began during the third week of January and is continuing. Following completion of mission sequence and environmental testing, the SC-5 spacecraft will undergo combined systems test on June 10, 1967.

SC-6 initial system checkout was rescheduled and should begin in the near future. Initial system checkout for the SC-7 spacecraft is scheduled for March 24.

Engineering was completed on the spherical auxiliary television mirrors to be incorporated on SC-5 through -7. The larger of the two mirrors has a 70-in. radius of curvature and the smaller mirror has a 25-in. radius of curvature. The 70-in. mirror is positioned to view the area of the crushable block and vernier engine 3. The 25-in. mirror may either view the area of the crushable block and thrust chamber 2 or be positioned to view the deployed position of the alpha-scattering experiment sensor head.

A study was completed to add a third spherical mirror on SC-5 through -7. This mirror would be mounted on the lower spaceframe member adjacent to fuel tank 1 and positioned so that its control ray of viewing is aimed at the center of the deployment area of the alpha-scattering sensor head on the lunar surface.

The vernier engine fuel lines for SC-5 through -7 were modified by adding a pressure transducer and three thermal sensors. This change was recommended by the *Surveyor II* failure review board.

4. Liftoff and Translation Maneuver

A study is being conducted to determine the allowable touchdown velocities for a lunar liftoff and translation maneuver. The flexible response computer program is being altered to correspond to various postlanding configurations of the antenna solar panel positioner. With this altered response program, a number of landing velocity conditions are being investigated to determine an envelope of lateral and longitudinal velocity combinations that do not cause any failures of the antenna solar panel positioner.

In the initial touchdown condition, the roll, elevation, and solar axes of the antenna solar panel positioner are all pin locked (the polar axis is self-locking). After touchdown, however, any combination of the three-pinned axes can be unlocked to provide favorable orientation of the antenna and solar panel. Once one of these axes is unlocked, it cannot be relocked because the locking pins are removed by firing a pyrotechnic squib. The only resistance to motion about an unlocked axis is provided by the detent function of the axis drive motors. If an applied moment about an unlocked axis exceeds the motor detent torque (taking into account gear ratios and gearbox efficiency), the motor will begin spinning and motion about the axis will continue until the load drops below the detent torque.

Uncontrolled motion about an unlocked axis, as described above, could cause a destructive impact between the solar panel or antenna and other spacecraft elements. Of particular concern is the situation where the solar panel could strike the mast, destroying solar cells and decreasing the power producing capabilities of the solar panel. Therefore, landing conditions for a liftoff and translation maneuver are being restricted to those that produce no moments about unlocked axes in excess of the drive motor detent torque.

C. Thermal Control

Solar-thermal-vacuum retesting of the SC-3 spacecraft was performed after its return from the Eastern Test Range. Initial evaluation of the SC-3 retest data indicated that spacecraft components in the region of leg 1 were running at temperatures noticeably above those observed during prior SC-3 testing at the same solar intensity. An environment shift appears to have occurred and an investigation has been initiated.

Using SC-3 retest data, a comparison was made between surface sampler flight hardware and thermal mockup data. The thermal effects of both flight hardware and mockups on the spacecraft were similar and no adverse effects were noted.

An investigation was conducted as a result of a decrease in the acceptable main battery temperature bandwidth at the time of radar altimeter and doppler velocity

sensor activation. Utilization of compartment A thermal control and heating assembly is recommended as the most efficient means of guaranteeing acceptable battery operating temperatures.

A review of the present nitrogen tank thermal design indicated that it will be inadequate for SC-5, -6 and -7. A new thermal design has been generated.

D. Power

Deletion of the auxiliary battery from the SC-5 and subsequent spacecraft has resulted in a potentially marginal voltage condition during the high power operations of terminal descent. A detailed load and critical voltage analysis indicated that a minimum main battery terminal voltage of 18.0 v is needed for proper system operation. During terminal descent, with the radar altimeter and doppler velocity sensor on, the current demand is approximately 45 amp (50 amp is the worst case). Added to this current demand is a 20-msec current pulse of approximately 19 amp at the time of retroengine firing. In SC-1 through -4 these currents were shared by the auxiliary and main battery on about a 50-50 basis. Thus there is practically no applicable test data available to establish whether the main battery alone can meet the new SC-5 requirement. Main battery data available from a reliability test program was evaluated and action was taken to obtain applicable test data. Several courses of action are being explored to ensure adequate main battery voltage during SC-5 terminal descent.

E. Propulsion

As a result of an action directive from the *Surveyor II* failure analysis review board, a test program was conducted to determine if the vernier engine thrust measuring accuracy could be improved. Since individual gages of the strain gage bridge were located in positions which could possibly subject them to different temperatures during vernier engine firing, the locations were changed in an attempt to remove any temperature sensitivity of the bridge.

Test firings of the vernier engine were made to compare the measuring accuracy between the original strain gage configuration, *Surveyor II* thrust data, and the relocated strain gage configuration. In addition, one test firing was made using an improvised radiation shield. The results of the tests indicated that the decay of thrust level was not affected by relocating the strain gages nor was the radiation shield effective after 6 sec of firing. It is therefore concluded that the decay rates are a result of thermal strain in the vernier engine upper mounting bracket, due to thermal gradients across the bracket. Further investigation of this problem is continuing.

F. Payload and Scientific Mechanisms

1. Alpha-Scattering Experiment

Schedules call for design, development, fabrication, and installation of the alpha-scattering-experiment hardware on the spacecraft in less than 5 mo. The major design objective is to reduce the risk to the mission created by changes to spacecraft configuration. Because of the potentially high risk associated with this program, a backup of mass and thermal equivalents is being developed. To ensure compatibility with the spacecraft, detailed interface analyses have been conducted, and the results have been verified through test.

During this reporting period, several significant test milestones were successfully completed: (1) the first flight-hardware subsystem integration tests in the payload systems laboratory; and (2) special dynamic tests of the alpha-scattering deployment mechanism and sensor head (Fig. 1) conducted on the S-9 structural test model.

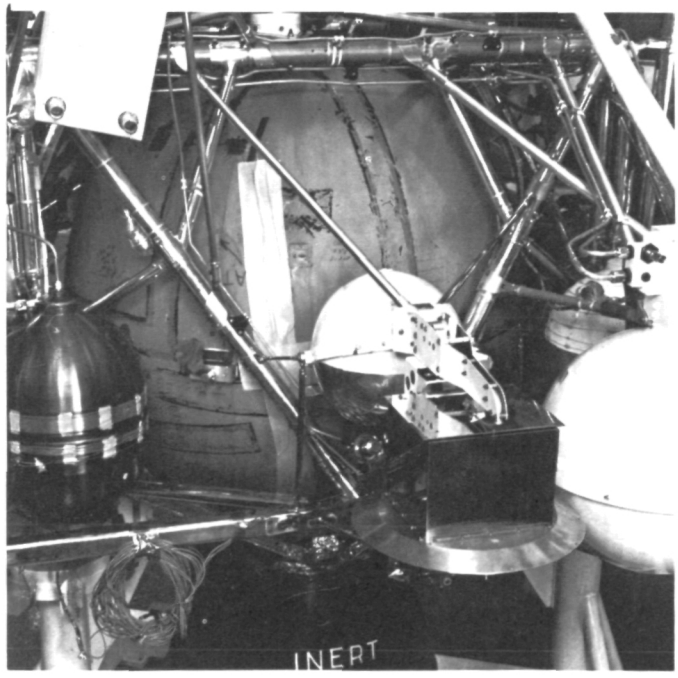

Fig. 1. Alpha-scattering sensor head and deployment mechanism installed on S-9 structural model

The subsystem integration tests demonstrated the electrical integrity of the alpha-scattering subsystem control items, the compatibility between each control item, and the compatibility of the subsystem with the spacecraft. The S-9 dynamic tests proved that the alpha-scattering deployment mechanism could withstand the vibration levels it will be subjected to during launch.

2. Surface Sampler

A basic objective of the surface sampler design and development program was to ensure that the surface sampler operation did not degrade overall spacecraft performance during test, flight, and lunar-surface operations. The design approach included detailed analyses of all critical circuits and spacecraft interfaces, and verification of each of the analyses through either subsystem or system testing, where practicable.

Electrical interconnections between the surface sampler subsystem and the spacecraft were established through the existing TV camera 4 harness. No spacecraft electrical or cabling changes were required, but it was necessary to design and develop logic circuitry to receive the existing four command inputs and to provide the fifteen commands required for surface sampler operation. In addition, a squib firing circuit had to be developed to release the pin puller and deploy the surface sampler mechanism prior to lunar surface operations. These additional circuits were included as part of the surface sampler auxiliary.

A strong measure of basic protection related to susceptibility of the surface sampler to electromagnetic interference is provided by this design approach, since the surface sampler auxiliary logic circuitry must receive a correct sequence of twelve commands to fire the surface sampler deployment squib. Protection is also provided to the spacecraft power subsystem by including circuits within the auxiliary which limit the amount of current the surface sampler can draw from the 29-v bus and by fusing the 22-v power utilized to drive the deployment squib and the surface sampler mechanism motors and heater.

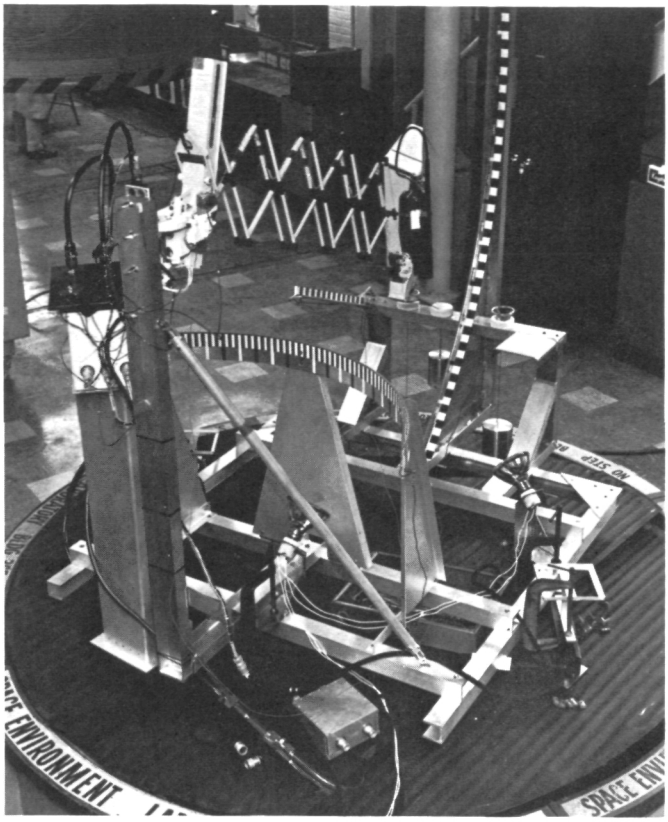

Fig. 2. Surface sampler checkout during SC-3 compatibility tests

In this reporting period, milestones were accomplished at the spacecraft level for both surface sampler 1 and 2. The most significant milestone was the troublefree and on-schedule installation and checkout of surface sampler 1 on the SC-3 spacecraft at the Eastern Test Range (Fig. 2). The second milestone was the successful completion of flight acceptance testing by surface sampler 2 and the troublefree installation of surface sampler 2 on SC-4. Compatibility testing of surface sampler 2 with the spacecraft is pending completion of SC-4 upgrade.

Additional testing of surface sampler 1, because of SC-3 retests, included a solar-thermal-vacuum test, a previbration test, and a postvibration test. These tests demonstrated that no surface-sampler-mechanism or spacecraft degradation has occurred due to surface sampler installation.

II. *Mariner* Venus 67 Project

THE PLANETARY–INTERPLANETARY PROGRAM

N67-27743

A. Introduction

The primary objective of the *Mariner* Venus 67 Project is to conduct a flyby mission to Venus in 1967 to obtain scientific information which will complement and extend the results obtained by *Mariner II* relevant to determining the origin and nature of Venus and its environment. Secondary objectives are to: (1) acquire engineering experience in the conversion of a spacecraft designed for a mission to Mars (spare flight spacecraft from *Mariner* Mars 1964 Project) into one designed for a mission to Venus and in the operation of such a spacecraft, and (2) obtain information on the interplanetary environment during a period of increasing solar activity. An *Atlas/Agena D* launch vehicle will be used.

Due to the minimum length of time (18 months) between authorization of the project and the launch opportunity, techniques and hardware developed during prior projects must be utilized to the fullest extent possible. The single flight spacecraft, designated M67-2, will be a converted *Mariner* Mars 1964 Project flight spare. Portions of the proof test model and certain critical spare units from the *Mariner* Mars 1964 Project are being prepared for use as a flight support spacecraft, designated M67-1. The flight support spacecraft will serve the double function of a pseudo-proof test model and a backup spacecraft for qualifying spare subsystems.

Various changes to the *Mariner* Mars 1964 spacecraft design are necessitated by the fact that the M67-2 flight spacecraft will travel toward, rather than away from, the Sun; also, conversions must be made to accommodate the revised encounter sequencing and science payload.

The S-band radio occultation experiment, one of seven scientific experiments approved for the mission, requires the use of only the RF transmission subsystem on the spacecraft. The celestial mechanics experiment uses only the tracking doppler data derived from the RF carrier. The ultraviolet photometer, helium magnetometer, solar plasma probe, and trapped radiation detector experiments are to be accomplished using existing instrumentation with only minor modifications. Only the dual-frequency radio propagation experiment requires the incorporation of a new scientific instrument into the payload. The Principal Scientific Investigators for these experiments are given in Table 1.

Table 1. *Mariner* Venus 67 Principal Scientific Investigators

Experiment	Principal Scientific Investigator	Affiliation
S-band radio occultation	A. J. Kliore	Jet Propulsion Laboratory
Ultraviolet photometer	C. A. Barth	University of Colorado
Dual-frequency radio propagation	V. R. Eshleman	Stanford University
Helium magnetometer	E. J. Smith	Jet Propulsion Laboratory
Solar plasma probe	H. S. Bridge	Massachusetts Institute of Technology
Trapped radiation detector	J. A. Van Allen	State University of Iowa
Celestial mechanics	J. D. Anderson	Jet Propulsion Laboratory

During this reporting period, the M67-1 spacecraft successfully completed major qualification testing. The first system verification test of the M67-2 spacecraft was made with only minor operational problems observed; the second system verification test is scheduled for early March.

B. Design and Development

1. Temperature Control Reference

The purpose of the temperature control reference is to compare preflight estimates of solar thermal input (solar intensity × absorptance) with the solar thermal input actually encountered in flight. The temperature control reference consists of three assemblies, one located at the tip of each of three solar panels. Each assembly (Fig. 1) is identical except for the coating used on the sunlit side and the sensor range selection. The three coatings are flat black and two different types of white. Each assembly consists of a 3-in.-diameter sensor area, a tee-section fiberglass epoxy laminate support, and an aluminum bracket which attaches to the attitude-control gas jet manifold at the tip of the solar panel. A standard flight transducer is mounted on the temperature-control-reference bracket to measure the temperature of the attitude-control manifold.

Each sensor, which is encapsulated in the center of the 3-in. diameter area at the outer end, is a platinum resistance thermometer consisting of three separate coils. The coils are used in combinations of series arrangements to provide the proper temperature range for the coatings used. Encapsulated in the center of the sensor area with the sensor is a 2-in. diameter copper mesh disc to provide temperature uniformity.

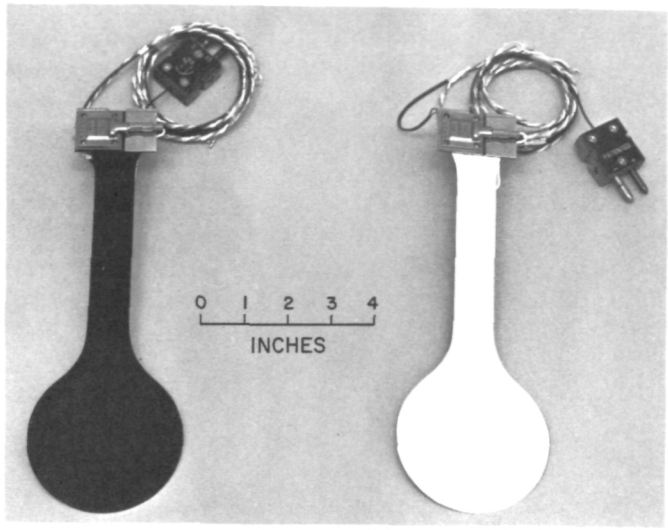

Fig. 1. Temperature control reference assemblies

2. Power Subsystem

Solar cell transducers similar to those flown on the *Mariner IV* solar panels are being provided for use on each panel of the *Mariner* Venus 67 spacecraft. These transducers consist of three special silicon solar cells which provide engineering measurements of short-circuit current and open-circuit voltage for correlation with similar measurements on the total solar panel array. The output of the transducer provides two short-circuit current measurements and one open-circuit voltage measurement.

The transducers were assembled, using identical materials and similar techniques to those of the respective solar arrays. Each of the cells was covered with a cutoff filter identical to that used on the solar panels. After assembly, all of the transducers were calibrated against balloon-flight standard cells.

C. Testing Equipment and Operations

1. Systems-Test-Complex Data System

Support of spacecraft system testing by the systems-test-complex data system continued during this reporting period. This included 250 hr of test support while the M67-1 spacecraft was in the JPL 10-ft space simulator.

During the continuous 176-hr Phase 1 portion of the test, the data system provided telemetry data processing, science data automation system (DAS) data processing (DAS hardline output, and output to Tracks 1 and 2 of the spacecraft tape recorder), and processing of the events and status data channels that had been checked out and were functioning at the time. Problems which occurred in the data system during the testing have been remedied by program or software system changes.

A real-time test support program with increased capabilities was assembled and is being tested during simulated countdown and launch tests of M67-2. A coding freeze will be established after demonstration of acceptable performance in full system test. It is not expected that any additional real-time data processing functions will then be implemented. An improved version of the non-real-time data processing program has been assembled and checked, and will be developed further to aid in reducing and preparing the recorded spacecraft test data for analysis.

2. Space Simulator Tests

The M67-1 spacecraft underwent testing in the JPL 10-ft space simulator. Two test sequences were performed; a system test (Phase 1) and a thermal test (Phase 2). Phase 1 of the test was intended to exercise the spacecraft system test configuration in a space environment. The Phase 2 objective was to verify the ability of the temperature control subsystem to maintain spacecraft temperatures within allowable bounds for all combinations of operating mode and flight environment. Secondary objectives were to check the accuracy of flight temperature transducers, to obtain more accurate flight temperature predictions, and to provide thermal support to the system test team in maintaining safe spacecraft and operational-support-equipment temperature levels. The thermal test was accomplished with minimum cables and operational support equipment in order to provide a flight-like thermal configuration. All test objectives were met. The test verified that the temperature control subsystem could maintain the spacecraft temperatures within allowable bounds for all combinations of operating mode and flight environment.

3. Attitude-Control Subsystem

Three problems occurred in the area of the attitude-control subsystem during the system testing phase. The chief problem was the failure of the pitch gyro in the gyro control assembly to maintain synchronous speed when the gyro control assembly was first installed on the M67-2 spacecraft. The gyro control assembly was removed from the spacecraft and the *Mariner* Mars 1964 proof-test-model gyro control assembly was installed so that testing could continue. A decision was made to contract for failure analysis of the pitch gyro plus two *Mariner* Mars 1966 gyros that had previously failed, and for repair or rebuild of at least two gyros. The goal was to have a flight-qualified gyro control assembly in the M67-2 spacecraft in time for environmental testing. The failure analysis has not been completed, but results so far indicate no evidence of a general failure trend that would affect all gyros.

Another problem was the apparent loss of torque gain of the jet vane actuators (which act in conjunction with the autopilot to control and orient the thrust vector of the midcourse engine). The designers of the actuators were consulted and the procedures for remagnetizing the actuator rotor, adjusting the rotor field strength, and measuring the torque gains were examined closely. New knowledge of the technique of adjusting the rotor field strength and measuring the torques was obtained, and the actuators that were suspected of losing torque gain were remeasured. The results indicated that the actuators were decreasing in rotor field strength and thus in torque gain; however, there was a limit on the decrease, and this limit was within the original torque gain specification for the actuators. Thus, the problem was resolved with no change to the actuators.

The third general problem was the discovery that the Canopus sensor was sensitive to magnetic fields applied along the axis of the image dissector tube that is used in the sensor. An analysis of the field of the undemagnetized sensor, as seen by the helium magnetometer, was undertaken. This analysis showed that the contribution from the sensor was small, so that it was acceptable to the magnetometer cognizant engineer, and that degaussing was not practical because degaussing would shift the null position on the photocathode, which would necessitate realigning the sensor optical assembly. In the future, construction techniques (involving less segmentation of the image dissector tube and ceramic seals, if necessary) will be investigated to minimize the problem.

III. *Mariner* Mars 1969 Project

THE PLANETARY–INTERPLANETARY PROGRAM

N67-27744

A. Introduction

The primary objective of the *Mariner* Mars 1969 Project is to conduct two flyby missions to Mars in 1969 to make exploratory investigations of the planet which will set the basis for future experiments – particularly those relevant to the search for extraterrestrial life. The secondary objective is to develop the technology needed for succeeding Mars missions.

The spacecraft design concept will be based on that of the successful *Mariner IV* spacecraft developed under the *Mariner* Mars 1964 Project. However, considerable modifications will be made to meet the 1969 mission requirements and to enhance mission reliability.

The launch vehicle will be the *Atlas/Centaur SLV-3C*. This vehicle, developed under contract for and direction of the Lewis Research Center by General Dynamics/Convair, has a single- or double-burn capability in its second stage and a considerably increased performance rating over the *Atlas D/Agena D* used in the *Mariner IV* mission.

Mariner Mars 1969 missions will be supported by the Eastern Test Range launch facilities at Cape Kennedy, the tracking and data acquisition facilities of the Deep Space Network, and other NASA facilities.

The six planetary-science experiments selected by NASA for the *Mariner* Mars 1969 missions are the following: TV, infrared spectrometer, ultraviolet airglow spectrometer, infrared radiometer, S-band occultation, and celestial mechanics. Additionally, a planetary-approach-guidance engineering experiment will be incorporated to test the feasibility and flight performance of onboard optical sensors and associated data processing techniques necessary for optical approach guidance. The Scientific Investigators for the planetary-science experiments are listed in Table 1.

During this reporting period, a number of configuration and design changes to the flight spacecraft were completed. System design of the spacecraft operational support equipment was begun. A series of subsystem preliminary design reviews was partly completed.

Table 1. *Mariner* Mars 1969 Scientific Investigators

Experiment	Scientific Investigator	Affiliation
TV	R. B. Leighton[a]	California Institute of Technology
	B. C. Murray	California Institute of Technology
	R. P. Sharp	California Institute of Technology
	N. H. Horowitz	California Institute of Technology
	J. D. Allen	Jet Propulsion Laboratory
	A. G. Herriman	Jet Propulsion Laboratory
	L. R. Malling	Massachusetts Institute of Technology
	R. K. Sloan	Jet Propulsion Laboratory
	N. Davies	Rand Corporation
	C. Leovy	Rand Corporation
Infrared spectrometer	G. C. Pimentel[a]	University of California, Berkeley
	K. C. Herr	University of California, Berkeley
Ultraviolet airglow spectrometer	C. A. Barth[a]	University of Colorado
	W. G. Fastie	Johns Hopkins University
Infrared radiometer	G. Neugebauer[a]	California Institute of Technology
	G. Munch	California Institute of Technology
	S. C. Chase	Santa Barbara Research Center
S-band occultation	A. J. Kliore[a]	Jet Propulsion Laboratory
	D. L. Cain	Jet Propulsion Laboratory
	G. S. Levy	Jet Propulsion Laboratory
Celestial mechanics	J. D. Anderson[a]	Jet Propulsion Laboratory

[a]Principal Scientific Investigator.

B. Design and Development

1. Spacecraft Mechanical Configuration

Since the last report on the *Mariner* Mars 1969 spacecraft configuration (SPS 37-41, Vol. VI), the spacecraft design has undergone considerable change. As of this reporting date, the spacecraft configuration (Fig. 1) is relatively firm. The salient aspects of this configuration are presented here.

a. Octagon. The octagonal main electronic compartment is the same size and shape as the *Mariner* Mars 1964 design. The attitude control/central computer and sequencer bay has been interchanged with the science bay to provide a more uniform distribution of dissipated power for temperature control purposes. Several bays have been repackaged to provide more efficient utilization of the available volume.

b. Solar panels. Four identical solar panels with a total area of 83 ft^2 are attached to the outboard corners of alternate bays of the octagon. During boost the panels are latched to each other through a spring/damper mechanism and a pyrotechnic pinpuller. After release of this latch, the panels are driven to a plane normal to the spacecraft z axis by deployment clock springs at the hinge line and latched to cruise dampers mounted on the outboard face of each solar panel bay.

c. Antennas. The radio frequency subsystem low- and high-gain antennas are mounted on the spacecraft in a manner similar to the *Mariner* Mars 1964 design. However, the antenna configurations are different. The low-gain antenna is longer, so that it extends above the solar panels in the boost configuration, and has a conical end to shape its pattern to the 1969 trajectory. The high-gain antenna is a circular parabolic dish, 40 in. in diameter. It is mounted in a fixed pointing position on a superstructure above the octagon.

d. Celestial sensors. The attitude-control primary celestial sensors are located on the top of the octagon. The primary Sun sensors, for pitch and yaw control, are mounted on pedestals in Bays IV and VIII to provide coverage of the forward hemisphere. The Canopus sensor, for roll control, is located above Bay VIII on the centerline of the bay. A large baffle is attached to the end of the Canopus sensor to minimize its sensitivity to stray light. The Sun gate is mounted above Bay IV, viewing parallel to the spacecraft z axis. The secondary Sun sensors, for coverage on the aft hemisphere, are configured in four assemblies located in Bays I, III, V, and VII on the bottom of the octagon.

e. Attitude-control gas assemblies. The gas assemblies, consisting of two mechanically interchangeable units, are mounted on the top of the octagon to provide a simple plumbing design and increased ease of installation and charging. Each gas assembly services gas jet manifolds on the tips of two adjacent solar panels.

f. Approach-guidance sensor. The approach-guidance planet sensor is mounted on the outboard corner of the octagon between Bays IV and V. The instrument is located so that it can view Mars for 10 days before encounter.

g. Scan platform. The two-degrees-of-freedom scan platform is mounted on the base of the spacecraft. The platform-pointed instruments are the infrared spectrometer and its cooling unit, ultraviolet spectrometer, wide- and narrow-angle television cameras, infrared radiometer, far-encounter sensor, and two narrow-angle

Mars gates. All the instruments are outboard of both platform gimbals, but are pointed in various directions to satisfy their particular objectives.

h. Spacecraft adapter and attachment/separation mechanism. The spacecraft attaches to the *Centaur* vehicle through a 21-in.-high conical adapter which contains a diaphragm at its base for separating the spacecraft environment from that of the *Centaur*. Additionally, the adapter provides pads and mounting locations for separation connectors and separation-activated devices, separation mechanisms, and cabling. The spacecraft attaches to the adapter through a V-band release mechanism which is a reapplication of the *Mariner* Mars 1964 device.

2. Spacecraft Temperature Control

a. Bus. The major features of the *Mariner* Mars 1969 thermal design for the bus are the same as for *Mariner* Mars 1964 and *Mariner* Venus 67. The bus is shielded, insofar as possible, from variable solar heating. Heat generated by the electronics is rejected mainly through variable emittance louvers located on all bays, except Bay II. The temperature differences between bays are minimized by providing good internal radiation and conduction heat paths.

b. Scan platform. The thermal design of the platform which orients the scientific instruments toward Mars is a significant departure from that flown on *Mariner* Mars 1964. The approach selected, in effect, creates a second bus which is thermally independent of the main equipment compartment. The scan platform (or science bus) will have a surface area of roughly 20 ft^2 which will be insulated as well as possible. This will minimize the amount of heat required to make up for unavoidable losses. Part of this power will be thermostatically controlled to compensate for the variations due to both uncertainties in the shielding and the switching on and off of the instruments.

During the encounter modes, electrical power is switched to the instruments and actuators on the scan platform. To prevent the instruments from becoming too cold during the cruise phases, heat is supplied to the platform. Some of this heat is thermostatically controlled electrical DC power and the balance is provided by a radioisotope, a shielded sintered pellet of promethium oxide.

c. Launch environment. The launch vehicle for *Mariner* Mars 1969 will be an *Atlas–Centaur*, which will be employed in a direct ascent trajectory. The ground cooling system will be the same as that used for the *Surveyor* launches. The ambient temperature within the shroud, prior to launch, will be maintained at 65–75°F (i.e., there will be no prechill as attempted on *Mariner* Mars 1964).

The thermal environment will be similar to that experienced in *Mariner* Mars 1964 except for two conditions which result from the change to the *Centaur* launch vehicle and the direct ascent mode. First, the rate of pressure decay during the ascent period will be roughly four times (60 torr/sec) that previously experienced. Care must be exercised in the detail design of the thermal blankets to assure adequate venting during this period. The second exception is the aerodynamic heating of the spacecraft after the separation of the nose fairing. An initial heating pulse is determined by the timing of the nose fairing jettison and, for some launch times, a second heating period is experienced as a characteristic of the direct ascent trajectory. A maximum heating parameter has been established for each of these conditions, and corresponding constraints on the ascent sequence and trajectories have been imposed. The initial heating pulse will not exceed 140 lb/ft sec (0.18 Btu/ft^2 sec) and the subsequent peak will not exceed 30 lb/ft sec (0.039 Btu/ft^2 sec). Under these conditions the upper shield will approach temperatures of approximately 300°F under the worse conditions.

3. Telemetry Subsystem

The telemetry and related subsystems were modified to accommodate three frequency-multiplexed channels and to replace the PN synchronization with bit synchronization carried out on the ground, using the telemetry and command processors at the Deep Space Instrumentation Facility stations. A science channel, operated during encounter and playback at 66⅔ or 266⅔ bits/sec, will contain scientific data from the data automation subsystem (DAS); an engineering channel, operated throughout the flight at 8⅓ or 33⅓ bits/sec, will carry engineering telemetry. A block-coded channel, operated at 16.2 kbits/sec, will provide playback or real-time DAS scientific data to the 210-ft-diameter advanced antenna system as appropriate during encounter and playback, alternating with the science channel.

4. Attitude-Control Subsystem

The attitude-control subsystem preliminary design review concentrated on redesign of the Canopus sensor and the gyro-control subassembly, repackaging of the

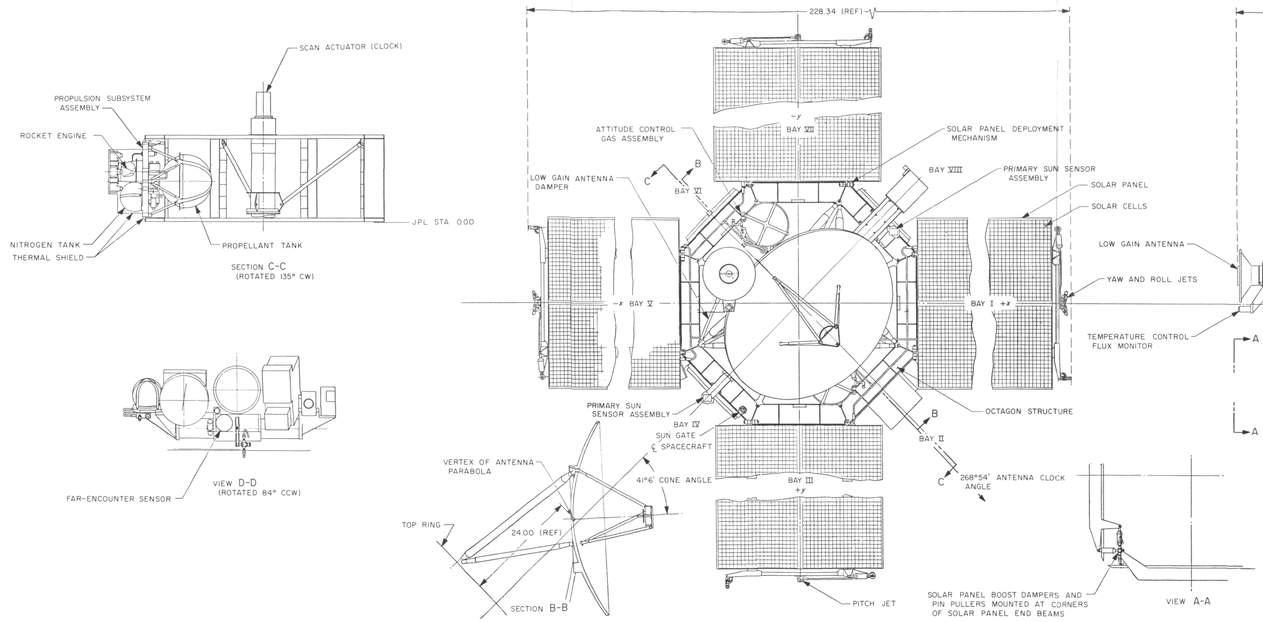

Fig. 1. *Mariner* Mars 1969 spacecraft mechanical configuration

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gyro-control subassembly and the attitude-control electronics subassembly, configuration changes on the gas system, and thrust-vector-control assembly design.

The major effort has been the redesign of the Canopus sensor. This redesign includes a roll plate loop mechanization, an adaptive low gate which periodically adjusts the brightness criteria for acquisition, a new Canopus sensor power logic which turns on Canopus sensor power with attitude-control turnon, and a new baffle and optical design to protect against stray light in the new location of the Canopus sensor.

Redesign of the gyro-control assembly includes a 0.4-in. increase of overall height, reorientation of the pitch and yaw gyro input axes, relocation of the elapsed time indicator, and increased alignment accuracy of the gyros.

5. Scan-Platform Subsystem

The scan-platform subsystem provides a mounting base for some of the scientific equipment to be used during encounter of the *Mariner* Mars 1969 spacecraft and the planet Mars. The platform is mechanized with two degrees of freedom, in clock, which is defined to be coincident with the spacecraft roll axis, and in cone, which is orthogonal to the roll axis. The scan-platform subsystem performance is divided into two modes: far-encounter, and near-encounter. The scan-platform subsystem is comprised of two gimbal servo drivers (cone and clock), four stepper motor-driven potentiometers (allowing for the prestorage of four discrete reference-angle positions), a far-encounter planet sensor, and mode-control electronics.

6. Propulsion Subsystem

Design changes in the propulsion subsystem, principally in the hydrazine reactor, were initiated so that Shell #405 spontaneous catalyst will replace the bipropellant start cartridge employed in *Mariner* Mars 1964. The Shell #405 catalyst spontaneously initiates hydrazine decomposition at propellant temperatures down to freezing, and at catalyst bed temperatures down to about 0°F. The redesigned propulsion subsystem is shown in Fig. 2.

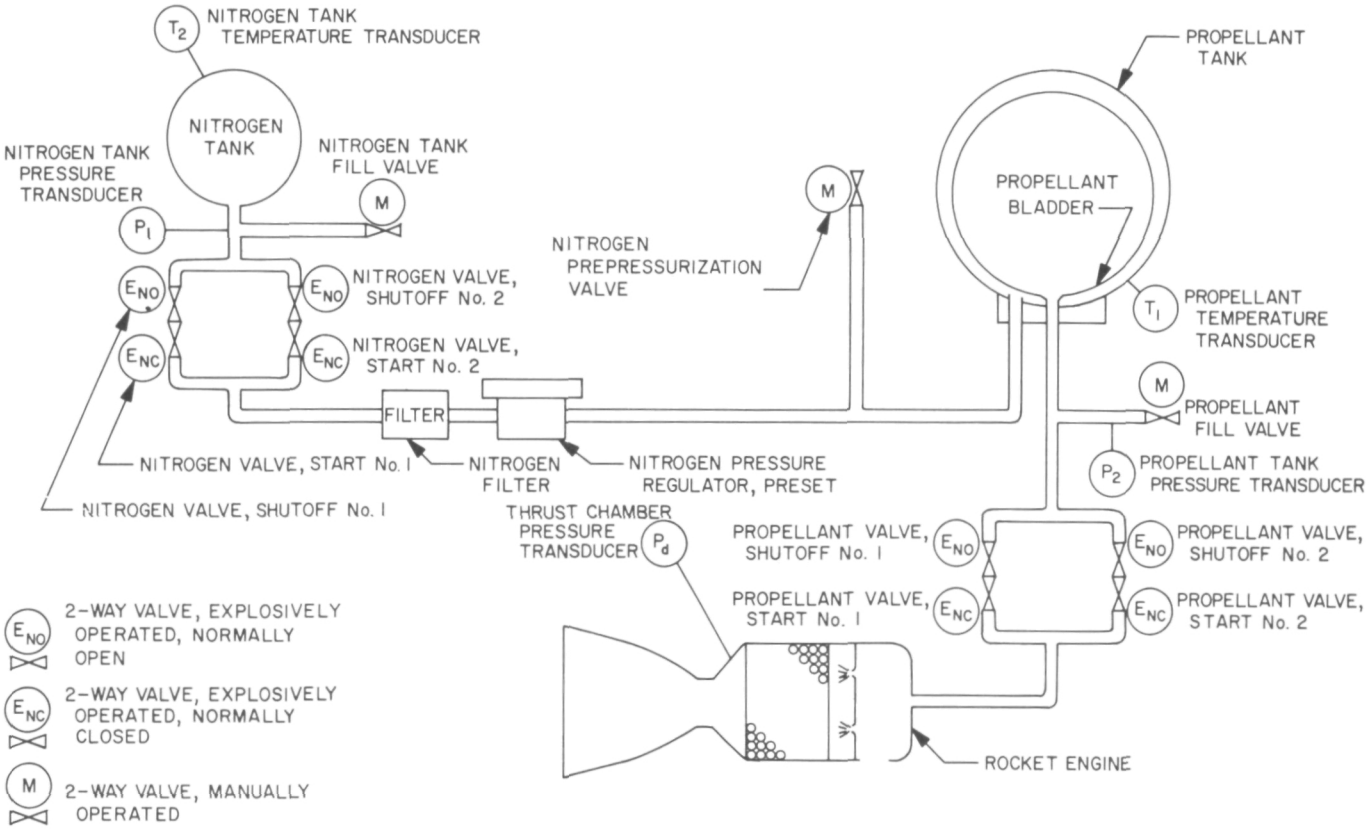

Fig. 2. Schematic of spontaneous-catalyst propulsion subsystem

The redesigned hydrazine reactor is shown in Fig. 3. The top 0.2 in. of the catalyst bed consists of 20–30 mesh Shell #405 catalyst and the lower portion consists of

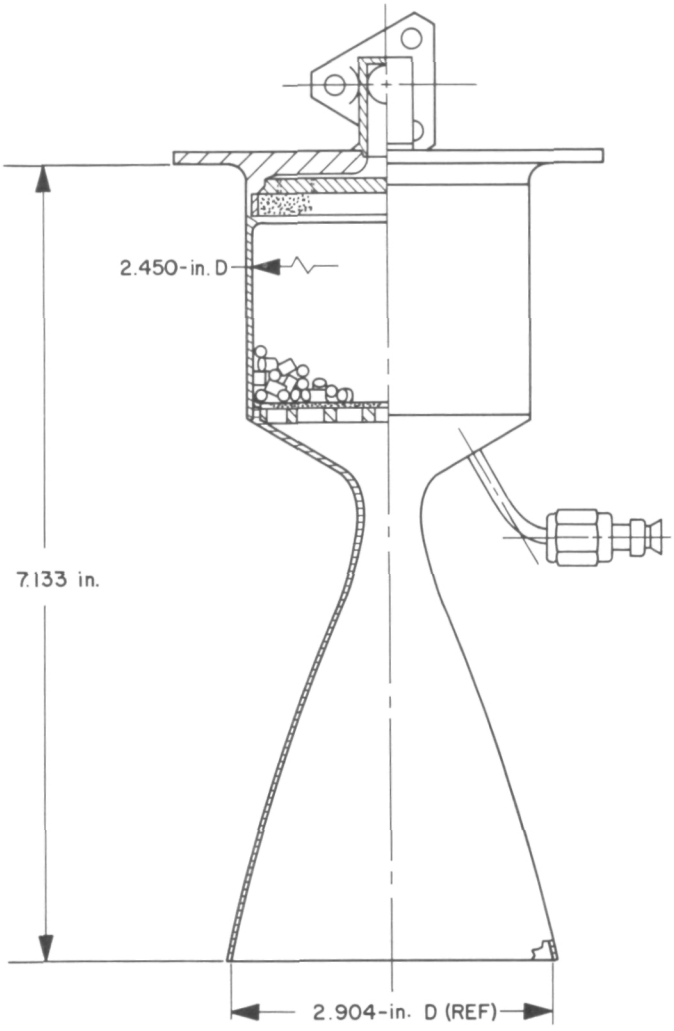

Fig. 3. Shell #405 catalyst, 50-lb thrust reactor

⅛-in. cylinders of the same catalyst. The injector of this engine consists of forty-two 0.017-in. diameter holes. The catalyst bed is slightly smaller in diameter than that of the *Mariner* Mars 1964 reactor and considerably shorter (about 2.2 in.). The fuel enters through a single inlet and is sprayed through the small holes onto the top of the catalyst bed. The same materials and construction techniques have been used as those used on the *Mariner* Mars 1964. An identical nozzle assembly is used in order that the jet vane interface will not be disturbed. The bolt circle at the top of the reactor also remains the same.

The spontaneous-catalyst engine will permit better predictability of the thrust time curve, since it approaches steady-state operation more rapidly than the existing engine. However, there will be a slightly lower performance capability (about 1% less). (Tests are being made to explore the possibility of packing a portion of the lower catalyst bed with the JPL Type H-7 catalyst to minimize the loss in performance.) In no case will the spontaneous-catalyst engine give a catastrophic failure. Experience indicates that if the catalyst or engine does not operate properly, the result is simply rough operation, or no operation at all. It is believed that the new engine will increase reliability greatly, but it has not been flight-proven. There is an estimated weight saving of 4.6 lb as a result of the new design.

7. Solar Panel Boost Dampers

Design of the solar panel boost dampers for *Mariner* Mars 1969 has remained essentially unchanged in concept from the *Mariner* Venus 67 design. The dampers provide the following functions: tip-to-tip latching of the solar panels in the stowed position; damping of the natural vibration modes of the solar panels; and support of the static loads associated with the solar panels. The damper consists of two concentric tubes and a coaxial

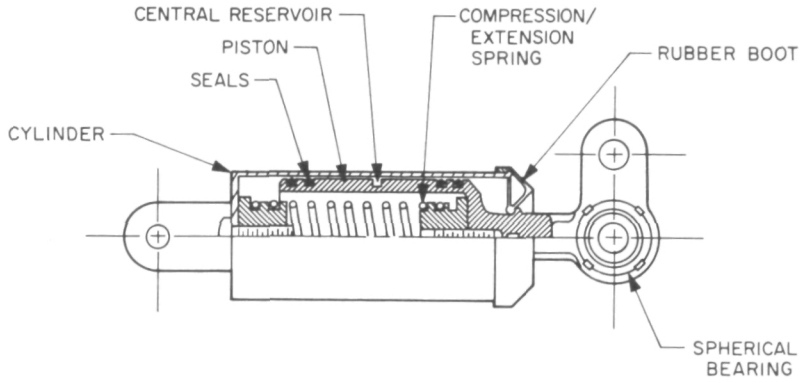

Fig. 4. Solar panel boost damper configuration

extension/compression spring, with a viscous damping fluid contained in the annulus by seals seated in the piston body. A central reservoir in the piston can supply fluid to the annulus to replace that which may have been removed by the wiping action of the seals. The damper configuration is shown in Fig. 4.

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IV. DSN Capabilities and Facilities

THE DEEP SPACE NETWORK

N67-27745

Established by the NASA Office of Tracking and Data Acquisition and under the system management and technical direction of JPL, the Deep Space Network (DSN) is responsible for two-way communications with unmanned spacecraft travelling from approximately 10,000 miles from the Earth to interplanetary distances. [Earth-orbiting scientific and communications satellites and the manned spacecraft of the *Gemini* and *Apollo* Projects are tracked by the Space Tracking and Data Acquisition Network (STADAN) and the Manned Space Flight Network (MSFN), respectively.] NASA space exploration projects supported, or to be supported, by the DSN include the following: (1) *Ranger, Surveyor, Mariner* Mars 1964, *Mariner IV, Mariner* Venus 67, *Mariner* Mars 1969, and *Voyager* Projects of JPL; (2) *Lunar Orbiter* Project of the Langley Research Center; (3) *Pioneer* Project of the Ames Research Center, and (4) *Apollo* Project of the Manned Spacecraft Center (as backup to MSFN).

Present DSN facilities permit simultaneous control of a newly launched spacecraft and a second spacecraft already in flight. In preparation for increased U.S. activities in space, a capability is being developed for simultaneous control of either two newly launched spacecraft plus two in flight, or four spacecraft in flight. Advanced communications techniques are being implemented to enable obtaining data from, and tracking spacecraft to, planets as far out in space as Jupiter. The main elements of the DSN are described below.

A. Deep Space Instrumentation Facility

The Deep Space Instrumentation Facility (DSIF) is composed of tracking and data acquisition stations around the world. The deep space stations (DSS's) of the DSIF and the deep space communication complexes (DSCC's) they comprise are as follows:

DSS	DSCC
Pioneer Echo Venus Mars	Goldstone
Woomera Tidbinbilla Booroomba[a]	Canberra
Johannesburg	
Robledo Cebreros Rio Cofio[a]	Madrid
Cape Kennedy (spacecraft monitoring)	
Ascension Island (spacecraft guidance and command)	

[a]Station not yet authorized.

These stations are situated such that three may be selected approximately 120 deg apart in longitude in order that a spacecraft in or near the ecliptic plane is always within the field of view of at least one of the selected ground antennas. JPL operates the U.S. stations and the Ascension Island DSS. The overseas stations are normally staffed and operated by government agencies of the respective countries, with the assistance of U.S. support personnel.

The Cape Kennedy DSS supports spacecraft final checkout prior to launch, verifies compatibility between the DSN and the flight spacecraft, measures spacecraft frequencies during countdown, and provides telemetry reception from liftoff to local horizon. The other stations obtain angular position, velocity (doppler), and distance (range) data for the spacecraft and provide command control to (uplink) and data reception from (downlink) the spacecraft. Large antennas, low-noise phase-lock receiving systems, and high-power transmitters are utilized. The 85-ft-diameter antennas have gains of 53 db at 2300 MHz, with a system temperature of 55°K, making possible significant data rates at distances as far as the planet Mars. To improve the data rate and distance capability, a 210-ft-diameter antenna was built at the Mars DSS, and two additional antennas of this size are planned for installation at overseas stations. In their present configuration, all stations except the Johannesburg DSS are full S-band stations. The Johannesburg DSS receiver has the capability for L- to S-band conversion.

It is the policy of the DSN to continuously conduct research and development of new components and systems and to engineer them into the network to maintain a state-of-the-art capability. Therefore, the Goldstone DSCC is also used for extensive investigation of space tracking and telecommunications techniques, establishment of DSIF/spacecraft compatibility, and development of new DSIF hardware and software. New DSIF equipment is installed and tested at the Goldstone DSCC before being accepted for system-wide integration into the DSIF. After acceptance for general use, the equipment is classed as Goldstone Duplicate Standard in order to standardize the design and operation of identical items throughout the system.

B. Ground Communication System

To enable communications between all elements of the DSN, the Ground Communication System (GCS) uses voice, teletype, and high-speed data circuits provided by the worldwide NASA Communications Network between each overseas deep space station, the Cape Kennedy DSS, and the Space Flight Operations Facility (SFOF, described below). The NASA Communications Network is a global network consisting of more than 100,000 route mi and 450,000 circuit mi interconnecting 89 stations, of which 34 are overseas in 18 foreign countries. Entirely operationally oriented, it is comprised of those circuits, terminals, and pieces of switching equipment interconnecting tracking and data acquisition stations with, for example, mission control, project control, and computing centers. Circuits used exclusively for administration purposes are not included.

Voice, teletype, high-speed data, and video circuits between the SFOF and the Goldstone DSCC are provided by a DSN microwave link.

C. Space Flight Operations Facility

During the support of a spacecraft, the entire DSN operation is controlled by the Space Flight Operations Facility (SFOF) at JPL. All spacecraft command, data processing, and data analysis can be accomplished within this facility. Operations control consoles, status and operations displays, computers, and data processing equipment are used for the analysis of spacecraft performance and space science experiments. Communications facilities are used to control space flight operations by generating trajectories and orbits and command and control data from tracking and telemetry data received from the DSIF in near-real time. The telemetry, tracking, command, and station performance data recorded by the DSIF are also reduced at the SFOF into engineering and scientific information for analysis and use by scientific experimenters and spacecraft engineers.

V. DSIF Development and Operations

THE DEEP SPACE NETWORK

N67-27746

A. Flight Project Support

1. *Surveyor* Project

System interface compatibility testing was conducted at the Pioneer DSS in preparation for the third *Surveyor* mission. Installation of the multiple-mission support equipment into the S-band system necessitated new interface testing. TV interface verification, the control data console command transmission test, and the word error rate test were also accomplished. Mission-independent equipment/control data console compatibility testing was continued.

2. *Lunar Orbiter* Project

Lunar Orbiter II spacecraft tracking was continued. Special tests were conducted at the Echo DSS with this spacecraft to obtain more information concerning dual tracking: (1) a spacecraft transponder oscillator drift characteristic test, and (2) spacecraft transponder interference and silent acquisition tests. Three passes of the spacecraft were used for Echo DSS operational training and testing, in preparation for the *Lunar Orbiter III* mission.

The *Lunar Orbiter III* spacecraft was launched from Cape Kennedy, Florida, on February 4, 1967. The Echo DSS acquired the vehicle at 14:23 GMT on February 5 at the start of the first pass. The initial lunar orbiting deboost was accomplished by a command from the Echo DSS on February 8. On February 14, the video test leader was read out at the Echo DSS.

During passes 5 and 7, *Lunar Orbiter III* was occulted by the Moon. For 40 min during the occultation, *Lunar Orbiter II* was turned on and successfully tracked. Tracking of *Lunar Orbiter II* continued on a special schedule basis through February 28.

3. *Pioneer* Project

On December 20, a special *Pioneer VI* tracking pass was conducted, using the 85-ft-diameter antenna at the Echo DSS. The purpose of this pass, which occurred at a time when *Pioneer VI* was at a local minimum distance from the Earth, was to determine if *Pioneer VI* could be tracked from a distance of approximately 129,000,000 km, and then to use these tracking data to update the *Pioneer VI* orbit. The more precise orbit determination

will be used as a basis for validating the *Pioneer VI* tracking data obtained by the Mars DSS 210-ft-diameter antenna. These data are being accumulated for support of a new celestial mechanics investigation. Tracking of the *Pioneer VI* spacecraft by the Mars DSS continues.

The *Pioneer VII* spacecraft continues to be tracked. The Echo DSS commitment with *Lunar Orbiter III* caused the prime tracking of the *Pioneer VII* to return to the *Pioneer* DSS on February 1. Multiple-mission support was provided from the Echo DSS *Pioneer* ground operational equipment and the telemetry and command processing equipment via microwave. A lunar occultation by *Pioneer VII* occurred during pass 156 on January 19, during which a special experiment was conducted by Stanford University experimenters. A precise orbit determination was accomplished, and the best estimate of the orbit and the orbit uncertainties were mapped to the time of lunar occultation to predict the uncertainties in the times of lunar encounter. Prime tracking of *Pioneer VII* was moved to the Mars DSS on February 20, with the Echo DSS providing multiple-mission support.

4. *Apollo* Project

A new *Lunar Orbiter* experiment is being planned, the primary purpose of which is the qualification of the *Apollo* Manned Space Flight Network (MSFN), including the unified S-band system, and the navigation subsystem of the Manned Spacecraft Center real-time computer complex at Houston, Texas. This experiment is divided into three phases:

(1) Phase A will include the collection and processing by the computer complex of two-way tracking data from the Deep Space Network (DSN) and three-way tracking from the MSFN. Three-way tracking by the MSFN will coincide with the scheduled tracking passes by the DSN.

(2) Phase B will include two-way tracking of *Lunar Orbiter* spacecraft by the MSFN; no ranging data will be obtained. During the first half of Phase B, three passes per month are to be tracked by part of the MSFN; during the second half, two passes per month are to be tracked by the entire MSFN.

(3) Phase C will include two-way doppler and ranging tracking data once per month by the entire MSFN.

During Phases B and C, the DSN will schedule three orbits of tracking data during each scheduled MSFN tracking pass. The first orbit will have overlapping coverage by two DSN stations (one in a two-way mode and the other station in a three-way mode) to determine a precise orbit for comparison purposes by the Manned Spacecraft Center computer complex. This precise orbit will also be used to satisfy some of the selenodesy tracking data requirements and the *Lunar Orbit* Project requirements for orbit updating, and may satisfy the requirements of other *Lunar Orbiter* tracking experiments.

A real challenge in qualifying the *Apollo* MSFN and the navigation subsystem is the requirement to evaluate the *Lunar Orbiter* thrust maneuvers by use of tracking and telemetry data.

B. DSS Equipment Installation and Testing

1. Pioneer DSS

Installation of two 500-kw diesel-engine generators is in progress. When completed, the installation will provide the Echo DSS with 1300 kw of back-up power. The S-band system has been augmented by an additional telemetry and command processing computer and an interim antenna pointing subsystem, which are being operationally tested.

2. Echo DSS

A 4-ft-diameter HA-Dec paraboloid antenna was installed at the Echo DSS. Using this antenna, time synchronization experiments were conducted between the Echo DSS and the Venus DSS 30-ft-diameter antenna via the inter-site passive microwave link.

3. Mars DSS

S-band system installation continued with most major elements of the subsystems operational. The rubidium frequency standard was installed, tested, and placed in operation, assuring the operations of a local prime frequency standard. The digital instrumentation subsystem communications buffer was installed and placed in operation. Installation of the telemetry and command data handling transfer racks is in progress in conjunction with the *Mariner* Venus 67, and subsequent multiple-mission projects. A 500-kw diesel-engine generator unit is being installed. This unit will bring the total available back-up diesel-electric power to 2000 kw.

The hydrostatic bearings that support the alidade structure are being aligned and tested. Testing of the

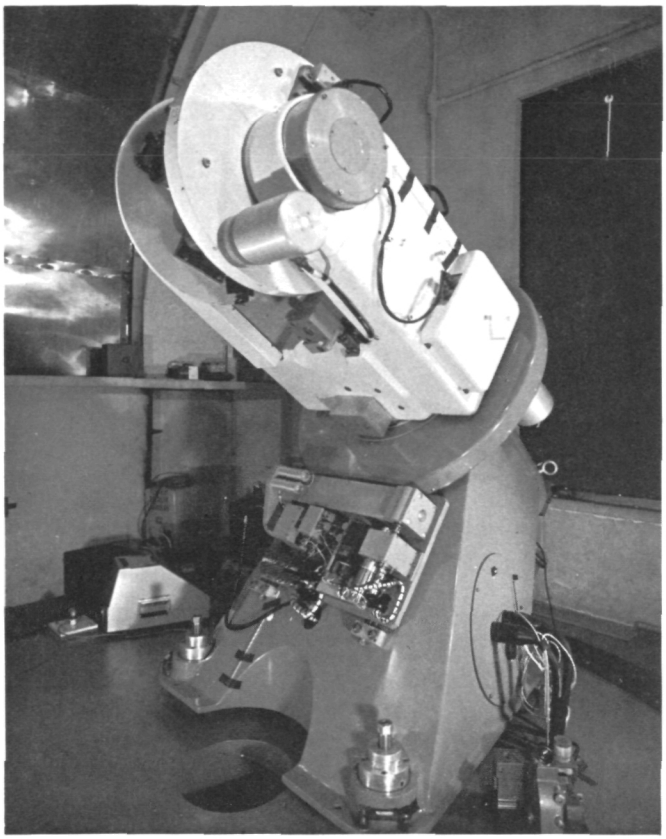

Fig. 1. Master equatorial installation at Mars DSS

master equatorial/intermediate reference structure antenna pointing continues. Located in an observation room at the top of the instrumentation tower, the master equatorial (Fig. 1) provides a precision angle data system for the 210-ft-diameter antenna.

C. Communications Development and Testing

1. Star Clock for Use in Antenna Calibrations

A star clock has been developed at the Tidbinbilla DSS to assist in selection of suitable stars for tracking (Fig. 2). This device is semiautomatic and includes a provision for the longitude and horizon masks of the observing station, which would be the only variables from station to station. The star clock has an outer local hour angle scale (and 24-hr time scale) containing the horizon mask fixed and an inner sidereal hour angle scale driven by a solar time clock mechanism 24-hr shaft. A small disc is driven by the 1-min shaft of the clock mechanism to provide an indication that the clock is operating. A manual adjusting knob is available in the clock mechanism for initial setting and adjustment of the inner, transparent scale. The star clock can perform several functions:

(1) It can display the stars within the field of view of the observing station (taking the horizon mask into account) and indicate the local hour angles of these stars to within about 1 deg, and maintain this accuracy for several hours.

(2) It can determine the rise and set times and meridian crossings of any star.

(3) It can determine the local hour angle of the Moon and planets within about 1 deg, and maintain this accuracy for a limited period.

(4) It can indicate sidereal time to an accuracy within a few minutes for several hours.

2. Step-Recovery Diode ×6 Frequency Multiplier

A step-recovery diode ×6 frequency multiplier has been developed to meet the requirements of the frequency synthesizer section of the *Mariner* Venus 67 precision ranging system. The output of this module is the coherent reference for the DSIF unified S-band RF receiver located at the Mars DSS.

The ×6 frequency multiplier converts a 10-MHz signal into a 60-MHz +13 dbm reference signal by means of a single-stage multiplier. This is accomplished by an input limiter/amplifier that excites a step-recovery diode. The multiplied signal is amplified and filtered by a crystal bandpass filter. An output amplifier restores the 60 MHz to the +13 dbm level required.

The advantages of this type of frequency multiplier are: (1) the high order of multiplication available with little loss, (2) the reduced number of electronic components required for high multiplication factors, (3) high power level outputs, (4) low power requirements, (5) smaller housing assembly, and (6) reduced sensitivity to temperature changes.

3. 100-kHz Square-Law Detector

One of the modules designed and built for the *Mariner* Venus 67 precision ranging system is a square-law detector. Its purpose is to convert a 100-kHz signal to dc, which is then fed into the digital section of the system.

A Hewlett-Packard crystal detector, Model 423A, was used as the square-law device because of its large dynamic operating range. In order for the device to operate linearly, a maximum of −3 dbm input power could be tolerated; this required a 16-db attenuator. The output

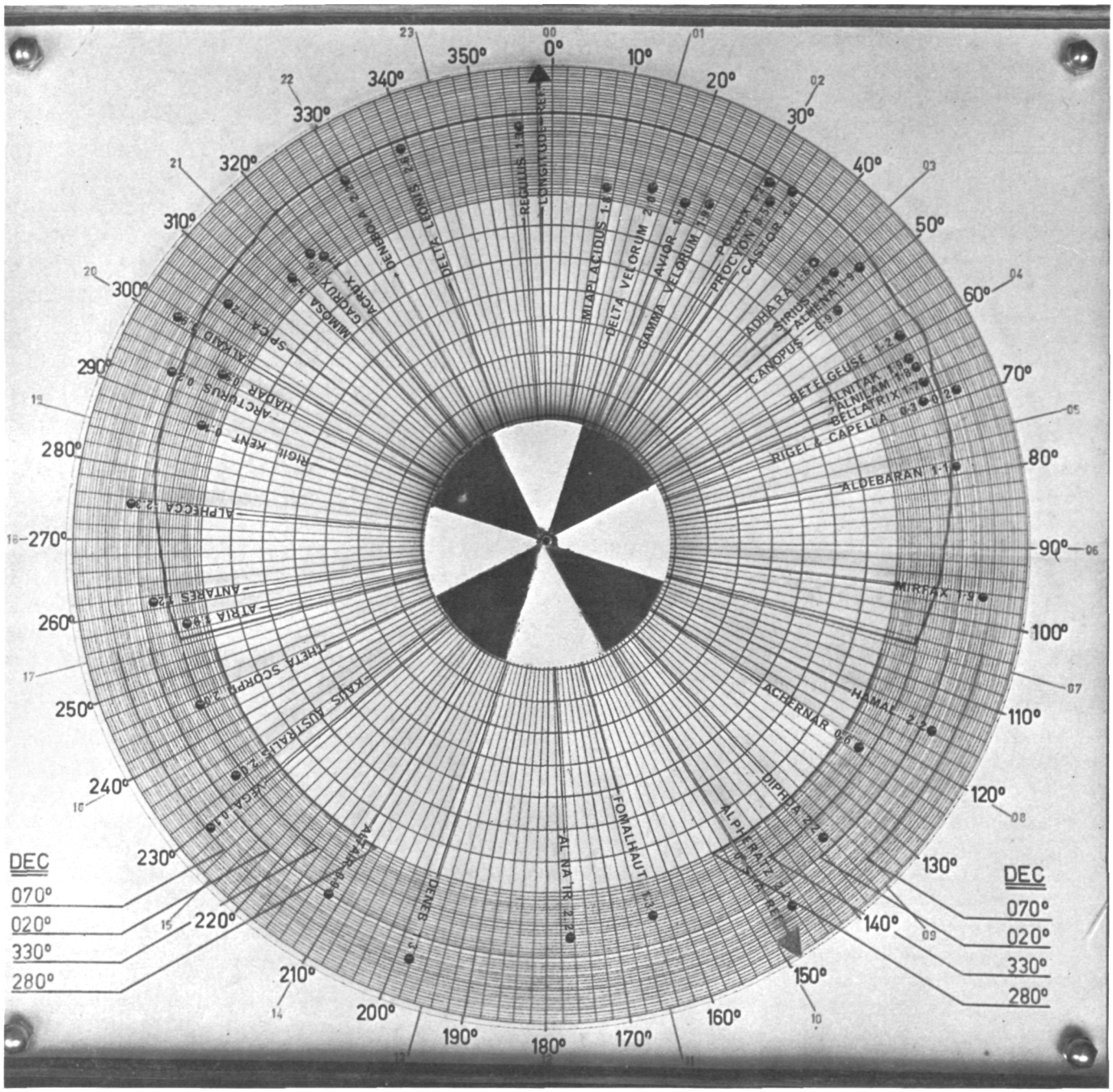

Fig. 2. Star clock (front face closeup)

of the detector is filtered by the RC circuit in the feedback loop of the operational amplifier. To minimize noise at the input of the operational amplifier, glass resistors were used along with a low-drift, low-noise operational amplifier. Current noise was minimized by using a circuit configuration which would allow smaller values of resistors to be used. An isolation resistor was placed at the output of the operational amplifier to suppress any parasitic oscillations which might occur due to the capacitive loading of the output.

4. 10.1-MHz Balanced Mixer

One of the modules that has been designed, built, and checked out for the *Mariner* Venus 67 precision ranging system is a 10.1-MHz balanced mixer which is used in the frequency synthesizer section to supply a 10.1-MHz reference signal.

The circuit for the 10.1-MHz balanced mixer consists of five sections: (1) a 10.0-MHz isolation stage, (2) a 0.1-MHz isolation stage, (3) a mixer stage, (4) a crystal filter, and (5) an output amplifier. The two input isolation stages provide proper isolation between the previous module and the mixer, ensure a low input voltage standing wave ratio, and regulate the amount of signal that is introduced into the mixer. The mixer stage consists of a Hewlett-Packard 10514A double balanced mixer with a 45-db typical rejection of the carrier. The next stage provides the filtering for the output of the mixer. The crystal filter has a 20-kHz bandwidth, with a 60-db rejection at ± 100 kHz either side of the 10.1-MHz center frequency. The last stage amplifies the signal to an output power of +13 dbm. A 2N918 transistor was used in the last stage because of its 10.1-MHz current gain as well as its low noise figure.

5. Traveling Wave Maser

Preparations for construction of a traveling wave maser with a tuning range suitable for operation between 2200 and 2300 MHz are being made. A 3-in. test unit has been fabricated and tested to provide necessary design data. Experimental results were compared with previously used traveling wave masers. Unfortunately, the configuration that provides highest gain degrades isolator performance and results in high forward loss. Single-crystal yttrium iron garnet (YIG) isolators were tested and found to be an improvement over the polycrystalline YIG isolators previously used. The single-crystal YIG isolator used in a comb configuration yields higher gain than can be obtained with previous designs.

The traveling wave maser will have a ruby-loaded comb structure. A gain adjustment trim coil will be used to trade excess gain for additional bandwidth. 30-MHz 3-db bandwidth and 20-MHz 1-db bandwidth will be available at a net gain of at least 35 db. The indicated tuning range will provide coverage of any 20-MHz portion (flat within 1 db) within the 2200- to 2300-MHz range. The 6°K maximum noise temperature is defined at the waveguide input to the maser package and applies to the 1-db bandwidth (center frequency ± 10 MHz).